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NUCLEAR ELECTRIC POWER FOR SPACE MISSIONS

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PREFACE

This paper was prepared for presentation at the Nuclear Power for Space session of the 29th Annual Meeting of the Institute of Aeronautical Sciences to be held in New York City on January 23-25, 1961.

NUCLEAR ELECTRIC POWER FOR SPACE MISSIONS*

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I. INTRODUCTION

The first portion of this paper covers present planning concerned with the spacecraft secondary power requirements for planetary, interplanetary, and lunar exploration. This planning and study is relatively independent of the means of getting the spacecraft to its destination, although electric propulsion must be included in the study to make it sufficiently complete for planning purposes. From the power requirement shown and the desired weights, the need of a nuclear power source is clearly indicated.

The second portion of the paper covers the use of electric propulsion for the final phases of propulsion to place the spacecraft at its destination. This portion attempts to show that there is more than just an improvement

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by the use of electric propulsion. There are certain missions where the need for electric propulsion is definite in that presently existing information shows it to be the only system capable of properly achieving the goals. Applying the same principles covered in the first portion of the paper on power and weight, there is no doubt that nuclear power is required.

II. SPACECRAFT SECONDARY POWER

The use of nuclear sources for spacecraft secondary power will be largely dependent upon the mutual compatibility of the nuclear sources with the various space missions and the corresponding vehicles. In addition, the nuclear systems will probably be required to demonstrate sufficient performance advantages over other power systems to warrant the increased hazards associated with their use.

The space missions planned for the next decade are indicated in Table I. Estimates of power system weights and power requirements for these missions have been included in order to indicate the approximate levels at which nuclear and "conventional" systems must compete. Values for the Surveyor and Prospector spacecraft are not indicated since these will be determined by the contractor.

The capabilities of various power sources are presented in Fig. 1, where weight is indicated as a function of power level. Probably the most significant power source parameter is the specific weight, or weight per unit power, since this determines to a large extent the performance of a

Table I. Power and weight estimates for space missions

Date	System	Vehicle	Mission	Raw power	Power system weight
1961	Ranger 1 and 2	Atlas-Agena	Lunar	160	256
1962	Ranger 3, 4, and 5	Atlas-Agena	Lunar	160	144
1962	Mariner A	Atlas-Centaur	Venus	300	210
1963-64	Mariner B	Atlas-Centaur	Mars	350	300
1963-65	Surveyor	Atlas-Centaur	Lunar	---	---
1965-70	Voyager	Saturn	Mars, Venus Mercury, Jupiter	2500	500-1000
1966-71	Prospector	Saturn	Lunar	---	---

system. Figure 2 shows the specific weights of the previously mentioned power sources as a function of power level. The values indicated for reactor sources do not include the weight of shielding; inclusion of shielding increases the specific weight by 20 to 100%, depending upon the radiation requirements. The figures shown for solar powered equipment do not, with the exception of the Sunflower system, include the weight of energy storage equipment, which may account for 10 to 50% of the weight of a system.

Examination of Fig. 1 indicates that on a specific weight basis the crossover point from solar photovoltaic panels to the reactor sources occurs at power levels of from 0.7 to 3 kw, corresponding to weights in the range from 400 to 600 lb. The crossover region indicated is based upon missions ranging from Mars to Venus; for missions further out than Mars the crossover point will occur at lower power levels. Note that the solar panel array which has been tentatively selected for Mariner B weighs about 200 lb, while the SNAP-10 reactor source, which has roughly the same power capability, weighs about 350 lb without shielding. The latter figure exceeds the present estimate of the total weight of the Mariner B power system, including batteries and converters. A likely alternative to the solar panels for Mariner B is a solar thermionic system, which holds promise of having a specific weight of about one-third that of the solar panels. However, a considerable amount of development remains to be accomplished before such a system can be demonstrated to have achieved satisfactory performance.

The radioisotope thermoelectric generators (RTG) appear to be best suited for use where a small amount of power for an extended period is required, particularly where solar energy is not available, or is available only on a relatively low-duty cycle. As an energy storage device the RTG greatly exceeds conventional batteries in performance; at a 15-watt level, for example, the corresponding performance figures for a curium-fueled unit are about 3600 watt hours per lb vs 80 watt hours per lb. As may be seen in Fig. 2, the RTG units do not appear suitable for large power levels due to their relatively high specific weight.

The advent of the Saturn booster and the corresponding capability of placing spacecraft containing power systems of 500 lb and more on trajectories to the Moon, Mars, and Venus appears to be the earliest time at which the use of nuclear reactors as a source for secondary power becomes advantageous. As with the radioisotope generator, the reactor systems appear particularly attractive for landing missions involving operation during the night portion of the cycle due to the absence of a requirement for energy storage. Missions to Mercury using advanced Saturn configurations also appear to be capable of making good use of a reactor for secondary power, although the weight margins are not very great.

The most advanced chemically fueled Saturn configuration presently planned does not appear capable of delivering a sufficiently large spacecraft to Jupiter to permit use of a reactor for secondary power. Solar power at Jupiter is not very attractive due to the relatively low solar flux level. Radioisotope sources appear feasible for this mission; however, due to the low power level and large distances involved, the communication bandwidths would be very low. It appears that a more exotic propulsion system, such as a reactor-powered ion engine, will be required in order to deliver a vehicle with sufficient secondary power capability to provide adequate communication bandwidths. The primary power capability of such a system would be much greater than that required to satisfy the needs of the secondary power system.

III. NUCLEAR ELECTRIC POWER FOR SPACE PROPULSION

To establish a need for electric propulsion, it is necessary to compare existing and planned systems. Such comparison is unfortunate because it implies competition. The objective of this presentation is not to establish a system competitive to chemical systems but to show that there are missions which can only be satisfactorily accomplished by electric propulsion. There are also other advantages that are not prime requirements and these will be covered lightly.

Electric propulsion has many applications in spacecraft, such as attitude control, orbital control, midcourse maneuvers, terminal maneuvers, etc. Since the studies for these uses are still in progress they will not be covered in this presentation. The electric propulsion covered includes only that propulsion necessary to take the spacecraft from an Earth orbit and to place it at its required destination. This destination may be a planetary fly-by, a planetary capture (arrive at the vicinity of the planet at the same time and with essentially the same velocity of the planet and therefore achieve some kind of planetary orbit), the planetary orbiter (whose altitude depends on the desired mission) or a combination orbiter and lander.

Since other factors are necessary to establish the "when" of development, no attempt will be made at this time to show a schedule of required development. Cost per pound of payload is a tremendous factor and until this can be analyzed and compared with possible development of other chemical systems a proper electric propulsion development schedule cannot be established.

A. CHEMICALLY PROPELLED SPACECRAFT

The optimum trajectories for all spacecraft vary with each mission, each few pounds of payload, every change of specific impulse, etc. Therefore, the performance figures included in this portion can be challenged, and errors possibly can be shown to exist. Yet, these figures are probably within a factor of 2 or 3 and are definitely within an order of magnitude of what can be achieved by these systems. Three chemically propelled booster vehicle systems are considered. These are either developed or under development with enough assurance that they will be used for space exploration. In order to present their capabilities in as brief a manner as possible, Table II shows the amount of payload that they can place into a 100 n. mi. Earth orbit.

Table II. Payload capabilities

Vehicle	Gross payload, lb (for 100 n. mi. orbit)
I	13,000
II	48,000
III	50,000

All three vehicles use liquid propellants and are staged. Vehicle II has four stages and a maximum of 29,000 lb fuel available in the orbital stage as part of the gross "payload" weight of 48,000 lb. Such limitation on the available fuel indicates that, to obtain adequate mass ratios on some chemical missions, the weight placed in orbit must be less than 48,000 lb. Extensive modification of computer trajectories may be necessary to eliminate these errors in payload weight calculation, for it would be obviously inefficient for a booster vehicle to put less than maximum weight into orbit. At present these modified computer studies have not been made, and resulting payload figures are probably on the low side.

Vehicle III is a five-stage vehicle. The fifth stage has not been specified yet, but the performance values used are an extrapolation of what could be achieved under the present development schedule.

Table III presents a portion of the studies and calculations made to date for the three vehicles. The figures shown are not necessarily optimum performance, as this depends on when the mission is to be flown and what

Table III. Payloads and flight times for chemical systems

Mission	Chemical system I		Chemical system II		Chemical system III	
	Time, days	Payload, lb	Time, days	Payload, lb	Time, days	Payload, lb
Venus Fly-by	87-133	1300	87-130	11200	-	-
	103-119	1600	103-119	12700	-	-
Venus Orbiter	-	-	87-130	1000	-	-
	-	-	103-119	1600-2300	-	-
Mars Fly-by	-	1650	-	-	-	11000
Mars Orbiter						
Elliptical	230	660	-	-	230	4700
Circular	260	320	-	-	230	3200
Mercury Fly-by	-	0	93	3500	96	3700
			97	4200	88	4100
Jupiter Fly-by	-	0	-	-	1000	6600
	-	-	-	-	500	3500
Saturn Fly-by	-	-	-	-	2190	4100
	-	-	-	-	1170	2500

the details of the mission are. They do bracket the capabilities of the three vehicles closely enough to establish at least an order-of-magnitude capability of payload weight.

There are several holes in this chart where data are not shown due to unavailability at this writing. The zeros shown for vehicle I performing the Mercury and Jupiter fly-by missions are true data since this system does not have the capability.

There are two additional missions this chart does not include. The first is a shot out of the plane of the ecliptic. Vehicle III is capable of placing approximately 12,000 lb at 11-deg inclination angle and 2500 lb at approximately 22-deg inclination angle. This vehicle is also capable of placing a 2500-lb payload to within 0.24 astronomical units from the Sun with a flight time of roughly 40 days. There are many configurations of boosters and stages that can be shuffled around and may possibly give better performance than that shown here. There is no implication that the figures show the maximum the chemical systems can do. These are figures that apply to the existing planned vehicles, and obviously when new systems are firmed up their capabilities must be determined.

B. ELECTRIC SYSTEMS

The electric systems considered in this study consist of no specific accelerator or thrust device. The specific impulses were those which the extrapolated state of the art indicated could, with reasonable assurance, be achieved (in some cases by more than one type of accelerator). These accelerators cover the various forms of ion or electrostatic propulsion, the electrothermo systems, and possibly the magnetohydrodynamic devices although, to repeat, the breakthrough for the production of flight equipment is not included in these estimates.

The electric powerplants represented here do not represent those being developed, planned, or even specified. They represent power levels and powerplant weights considered achievable at some future date. No scientific breakthrough is scheduled to accomplish these, although some engineering breakthroughs or inventions will undoubtedly be necessary before flight articles meeting these requirements are produced.

All of the spacecrafts considered here use chemical systems to place them into an Earth orbit. In other words, they are identical with the chemical systems, the difference being that the spacecraft includes a form of electric propulsion which is used to achieve Earth escape rather than relying on the chemical system. Table IV is a brief description of the boost vehicles considered. It shows the number of spacecraft pounds that can be placed into an approximate 300 n. mi. orbit. There is no reason, other than an intuitive feeling, that 300 n. mi. is established as the orbit. As the studies continue, undoubtedly a different orbit for the startup of electric propulsion could be used. This may be selected for safety or other known performance reasons.

Table IV. Payload capabilities

Boost vehicle	Gross payload, lb (for 300 n. mi. orbit)
I	9,000
II	15,000
III	45,000

Boost vehicle II is a modified or more advanced version of the chemical vehicle I, shown in Table II, and the boost vehicle I shown in Table IV.

Boost vehicle III of Table IV corresponds to the chemical vehicle II shown in Table II. There is a deliberate attempt to use the smaller boost vehicles with the electric propelled spacecraft to further emphasize the ability to put up larger payloads. This also may have an effect on the cost per pound of payload.

Four powerplants were used in analyzing the capabilities of these systems. They do not necessarily represent specific powerplants, yet the power outputs and weights are taken from development or anticipated future development as shown in Table V.

Table V. Powerplant comparisons

Powerplant	Output	Pounds per kw
A	60	50
B	300	10
C	1,000	10
D	10,000	1

Note that lifetime is not considered in this comparison, even though when examining flight times it will be noted that we are talking of powerplant life expectancy of at least 10, 000 hr and approaching 50, 000 hr in the extreme missions.

Table VI shows (within a factor of 2 or 3) the range of payload weights and flight times for the planetary missions. This table includes only powerplant A and boost vehicles I and II. Table VII shows a similar chart for powerplant B and the three boost vehicles described. Table VIII shows powerplants C and D with boost vehicle III.

There are many omissions in these tables where data are being generated. With the limited manpower and to minimize computer time, there is a tendency to obtain only those data that we feel are significant. By a little extrapolation most of the other weights and flight times can be reached closely enough to establish whether there is a need for that specific mission.

For an out-of-the-ecliptic shot vehicle I and powerplant A can achieve approximately 15-deg inclination. Vehicle I with powerplant B can achieve approximately 40-deg inclination angle with around 3000-lb payload. A solar probe has not been thoroughly investigated due to the fact that the initial powerplant design has sufficient temperature problems without the addition of getting closer to the Sun. These data will be generated, though, sometime in the future.

Some studies have been made concerning use of electric propulsion for lunar missions. If time is not important, vehicle I can place 3000 lb into

Table VI. Payloads and flight times for powerplant A

Mission	Boost vehicle I		Boost vehicle II	
	Time, days	Payload, lb	Time, days	Payload, lb
Venus Capture	240	1850	-	-
	380	4200	-	-
Venus Orbiter (500 n. mi. final orbit)	-	-	-	-
Mars Capture	300	1500	285	9000
	400	3200	390	12000
Mars Orbiter (500 n. mi. final orbit)	320	800	-	
	400	2250	-	-
Jupiter Fly-by	630	900*	-	-
	850	1800	-	-
Jupiter Capture	-	-	850	6000
	-	-	600	2500**
Mercury Capture	-	-	700	5000
	-	-	280	2500**

*Will escape solar system
**Electrothermal (arc jet) accelerator

Table VII. Payloads and flight times for powerplant B

Mission	Boost vehicle I		Boost vehicle II		Boost vehicle III	
	Time, days	Payload, lb	Time, days	Payload, lb	Time, days	Payload, lb
Venus capture	-	-	-	-	240	21000
	-	-	-	-	360	33000
Venus orbiter (500 n. mi. orbit)	-	-	-	-	-	-
Mars capture	-	-	-	-	-	-
Mars orbiter (500 n. mi. final orbit)	100	2600	180	4000	-	-
	320	5000	360	11500	-	-
Jupiter fly-by	-	-	400	7000	-	-
	-	-	600	10500	-	-
Jupiter capture	-	-	-	-	290	12500
Mercury capture	-	-	-	-	400	20500

Table VIII. Payloads and flight times for powerplants C and D

Mission	Boost vehicle III		Boost vehicle III	
	Time, days	Payload, lb	Time, days	Payload, lb
Venus capture	160	21000	-	-
	200	28000	-	-
Venus orbiter	-	-	-	-
Mars orbiter	180	4000	-	-
	400	28000	-	-
Jupiter fly-by	500	18000	-	-
	800	27500	-	-
Jupiter capture	500	14000	180	9000
	850	27000	400	25000
Mercury capture	160	10000	-	-
	260	24000	-	-
Saturn capture	730	12500	250	4000
	970	21000	550	26000

a 94-mile lunar orbit in approximately 100 days. If a slow "freight" is permitted, 5000 lb may be placed into a lunar orbit in about 300 days.

C. SCIENTIFIC MISSIONS

In evaluating the use of electrically propelled spacecraft the space science implications were investigated. The purpose was to determine what problems might exist and what advantages might exist using such a spacecraft. Four missions were considered. These missions consist of a Mars satellite, a solar probe, a Jupiter probe, and an out-of-the-ecliptic probe.

Using electrically propelled spacecraft for a Mars satellite as compared to a chemical system, it was determined that the slow spiral into the selected orbit provided an excellent opportunity to obtain complete and quite accurate values for any radiation belt such as the Van Allen belt, which might exist at Mars. This slow spiralling should also enhance our ability to obtain the maximum information on the composition and structure of the Martian atmosphere.

There is no need to go into the details of the interest associated with a solar probe. Because of the temperature problem associated with such a probe, coupled with the anticipated problems of the power conversion systems in an electrically propelled spacecraft, it was decided that this should be postponed until a later date.

The Jupiter probe, with electric propulsion, is of particular interest principally because of the timesaving involved. The electrically propelled spacecraft will actually reach Jupiter more quickly than the chemically propelled system and with the 1-mw powerplant has a payload large enough to make many significant measurements. Since Jupiter is the closest of the major planets (that is, those planets which differ radically from the Earth, Mars, Venus, and Mercury, both in composition and structure), it offers many exciting possibilities. Photographic investigation of the Jupiter satellites for comparison to our Moon would produce some exceptional data. Radio astronomy has aroused much curiosity about the temperature and radiation belt trapped in Jupiter's magnetic field. Measurements of this magnetic field, etc., will be a great stride forward.

The out-of-the-ecliptic space probe is necessary to establish more data on such questions as, "What is the rate of mass loss of the Sun and other stars similar to it?" If the probe can be placed far enough out of the ecliptic, even a photograph of our own solar system should produce important data.

The significant feature of this investigation is that it produced advantages for the use of electric propulsion systems for all but the solar probe. The advantages actually consisted of the time to spiral into planetary orbit plus the available power for scientific instruments and communications after the destination was achieved. This implies the powerplant should have a lifetime in excess of that required for propulsion. Due to the flight time shown in the previous figures the additional 60 to 100 days should not be a heavy restriction to place on these powerplants.

IV. CONCLUSIONS

It is unnecessary to dwell on the nuclear powerplant extensively in the propulsion portion of this paper. When we speak of 50 lb per kw or less, it appears that a nuclear source is the only one available at these power levels with the required lifetime expectation. The power conversion systems are subject to much discussion, and range from the turboelectric to direct conversion in the reactor. The low weight per kilowatt shown in the Figures and Tables will be achieved by furthering the state of the art in the conversion field.

Table IX shows a gross comparison between those chemical systems included in this paper and the electrically propelled systems. Where the electric system is unique it has been underlined on the Table. There are other areas on the Table where the electric system appears to give a significant advantage, but, as stated previously, other factors must be considered.

Certain engineering judgment will consider required lifetime for these missions as being unachievable for some years. Lifetime is certainly one of the big problems associated with the use of electric power in space. Yet, with straight chemical systems a Jupiter or Saturn probe requires even longer life for the payload and its transmitter than does an electrically propelled spacecraft.

Table IX. System comparison

System	Mars capture		Mars orbit		Jupiter probe		Mercury probe	
	Days	Pounds	Days	Pounds	Days	Pounds	Days	Pounds
Chemical I	230	660	260	320	-	0	-	0
Centaur Boost vehicle I								
60 KW SNAP-8 Powerplant A	400	3200	400	2250	630	1500	-	-
Saturn C1 Boost vehicle II								
60 KW SNAP 8 Powerplant A	390	<u>12000</u>	-	-	850	6000*	400	<u>5000</u>*
300 KW Spur Powerplant B	-	-	360	<u>11500</u>	<u>600</u>	<u>10500</u>	-	-
Chemical III	230	4700	230	3200	1000	6600	88	4100
Saturn C-2 Boost Vehicle III								
Spur Powerplant B	-	-	-	-	-	-	400	20500*
1 meg Powerplant C	-	-	400	<u>28000</u>	850	<u>27000</u>*	260	<u>24000</u>*

*Planet capture (elliptical orbit)

An additional advantage that electrical propulsion has is the opening up of the "firing window"that presently exists for some of our planetary probes. The payload curve is very steep in relation to firing time and at present the window is measured in days. With electric propulsion it is conceivable that this firing window may be opened up to as much as three weeks. We expect that the effective exploration of space definitely requires nuclear electric propulsion. To obtain the scientific data we are all seeking, we will automatically use such systems. The research and the development necessary to bring about the application of this power to space vehicles must be encouraged.

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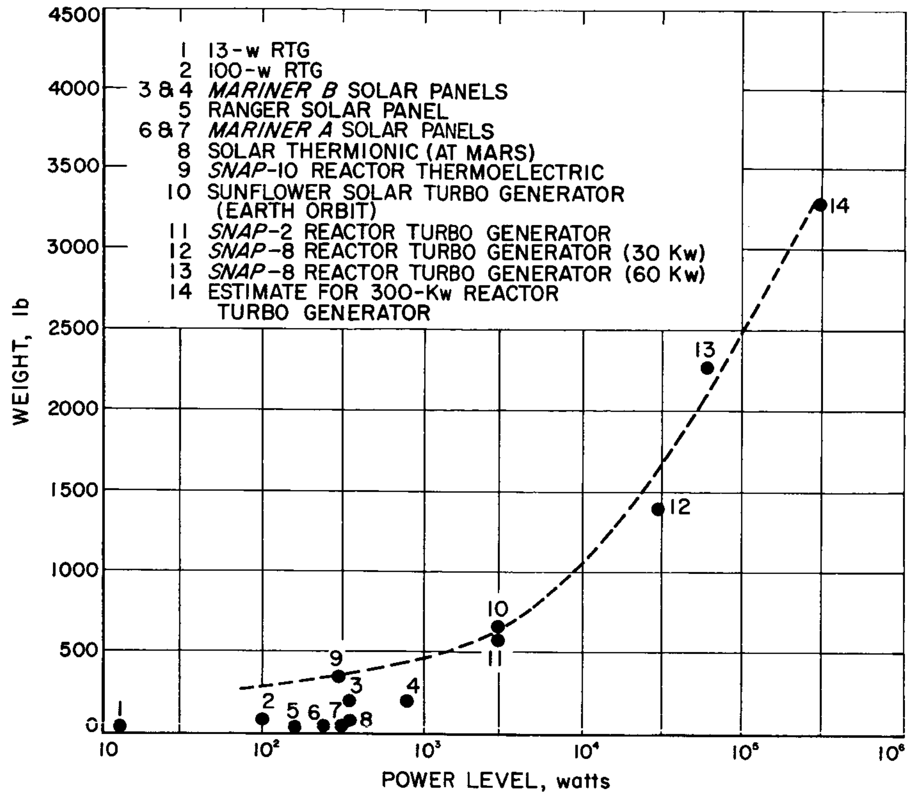

Fig. 1. Weights of power sources vs power levels

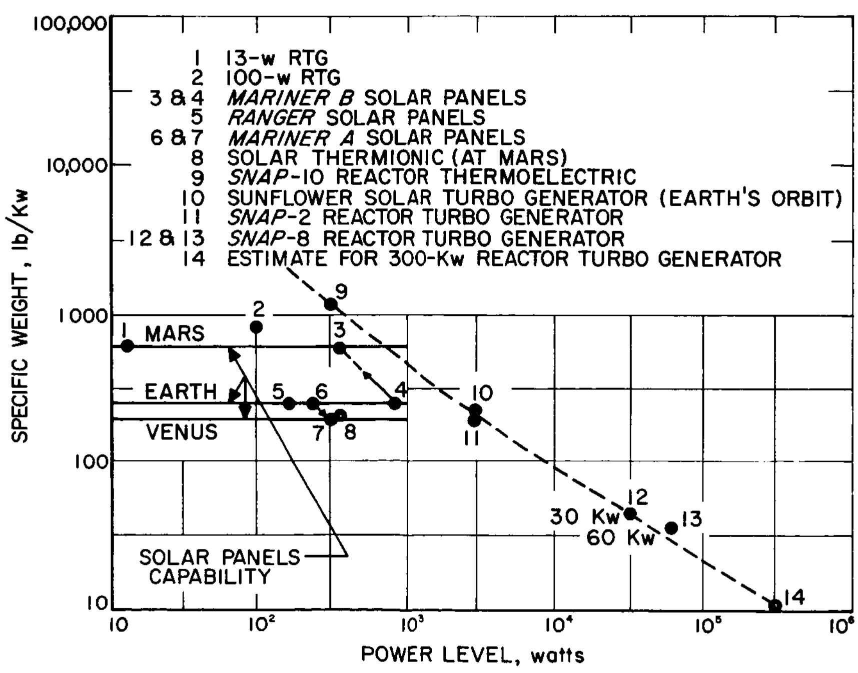

Fig. 2. Specific weights of power sources vs power levels